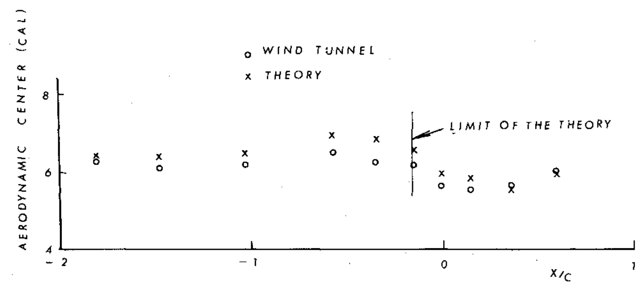

Fig. 4 Aerodynamic center of the store.

cause compressibility effects are minor[4]—a partial justification of slender body theory. A comparison of the experimental and theoretical vectors (as illustrated) suggests that the theory is sufficiently accurate for preliminary design purposes.

The pitching moment coefficient C_m and yawing moment coefficient C_n are similarly combined into a single vector $(C_m, -C_n)$. Define the "aerodynamic center" as that moment reference point on the axis of the store for which the resultant vectorial moment has a zero component along the cross-force vector. With this definition, the position of the aerodynamic center from the nose of the store is shown plotted in Fig. 4 for different (longitudinal) store locations.

The calculations are logically acceptable only when the trailing edge of the store is forward of the trailing edge of the wing ($x/c \leq 0.15$). However, if the calculations are continued aft of this point, the agreement appears quite satisfactory. This is so despite the disparity (Fig. 2) in the local flowfield. It is possible that the integrated effect of flow angularity is measured more adequately by the aerodynamic center of a store serving as a probe than by a rake.

References

[1] Adams, M. C. and Sears, W. R., "Slender-Body Theory—Review and Extension," *Journal of Aeronautical Sciences*, Vol. 20, Feb. 1953, pp. 85–98.

[2] Alford, W. J. and King, T. J., "Experimental Investigation of Flow Fields at Zero Sideslip, Near Swept- and Unswept-Wing-Fuselage Combinations at Low Speed," RM L56J19, Jan. 1957, NACA.

[3] Alford, W. J., "Experimental Static and Aerodynamic Forces During Simulated Launching from the Midsemispan and Wing-Tip Locations of a 45° Sweptback Wing-Fuselage Combination," RM L55A12, April 1955, NACA.

[4] Alford, W. J. and King, T. J., "Experimental Static Aerodynamic Forces and Moments at High Subsonic Speeds on a Missile Model During Simulated Launching from the Midsemispan Location of a 45° Sweptback Wing-Fuselage-Pylon Combination," RM L56J05, Jan. 1957, NACA.

Toward Simpler Prediction of Transonic Airfoil Lift, Drag, and Moment

Frederick O. Smetana* and Donald P. Knepper†

North Carolina State University, Raleigh, N.C.

DESIGN of wings for operation at transonic speeds has been hampered by the inability to predict their aerodynamic characteristics accurately. The difficulties stem, as is widely known, from the inherent nonlinearity present in inviscid treatments,[1] the significant influence of viscosity in the flow outside the boundary layer, and coupling between the boundary-layer flow and the external flow. Attacks on the problem have usually followed one of two courses: Solve the inviscid problem by various numerical techniques[2] or employ semiempirical correlations[3] of experimental data.

Because experiments carried out in our laboratory convinced us that viscosity has effects on the pressure distribution even at low angles of attack that could not be ignored and because the results obtained with many entirely analytical methods are not as accurate as required for satisfactory design work and are, in addition, expensive to use, we chose the second course. Following this approach one has, of course, almost an infinite variety of data correlations and theoretical calculations to choose from in assembling a method to predict aerodynamic characteristics of an airfoil between $M = 0$ and $M = 1.0$. To test the reliability of the approach selected, only simple versions of its analytical portions were used. The results have been sufficiently promising that more accurate versions are now being incorporated into the method. The method consists of five steps:

1) Compute the $M = 0$ pressure distribution using a distribution of vortices along the chord.[4] Use the Karman-Tsien method to correct for changes in Mach number up to the critical. A more accurate version of this procedure[5] includes boundary layer displacement effects.

2) Compute the $M = 1.0$ pressure distribution using the empirical technique of Thompson and Wilby[2] or the analytical technique of Truitt[6] if $\alpha = 0$.

3) Compute the pressure distribution rearward from the airfoil crest for $M_{CR} \leq M \leq M = 1$ using the semi-empirical technique of Sinnott and Osborne.[7] Fitzhugh[3] also makes use of this technique. In the region between the sonic line and the shock it is reasonable to expect that ultimately a completely analytical technique could be developed to account for the wave interactions and thus to duplicate the results of Sinnott and Osborne.

4) Assume the pressure variation over the first 5% of the airfoil is constant for $M \geq M_{CR}$. Then use a cubic spline to represent the pressure distribution between $x/c = 0.05$ and the crest. The fit should match the value and slope of the pressure distribution at the airfoil crest as well as the pressure value at $x/c = 0.05$.

5) Compute the Reynolds Number at the shock. For laminar boundary layers assume the pressure rise through the shock begins 50δ upstream of the predicted shock location. The pressure then rises linearly to the value downstream of the shock at the predicted shock location.

$$\delta = 5.73x/(Re_x)^{1/2}$$

For turbulent boundary layers the pressure rise extends linearly over 5δ turbulent. Obviously, this is a very crude approximation. More accurate versions are given in the papers by Murphy and by Rose.[8]

The procedure represented by the five steps was programmed for solution on a digital computer. Complete distribution could be obtained for 10 Mach numbers at one angle of attack in 25 seconds on an IBM 370/165. The

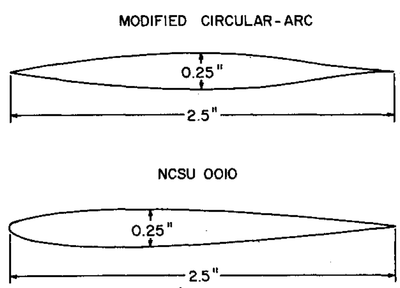

Fig. 1 Airfoil sections tested.

Received October 2, 1972.

Index categories: Airplane and Component Aerodynamics; Aircraft Configuration Design; Subsonic and Transonic Flow.

*Professor of Mechanical and Aerospace Engineering. Member AIAA.

†Research Assistant.

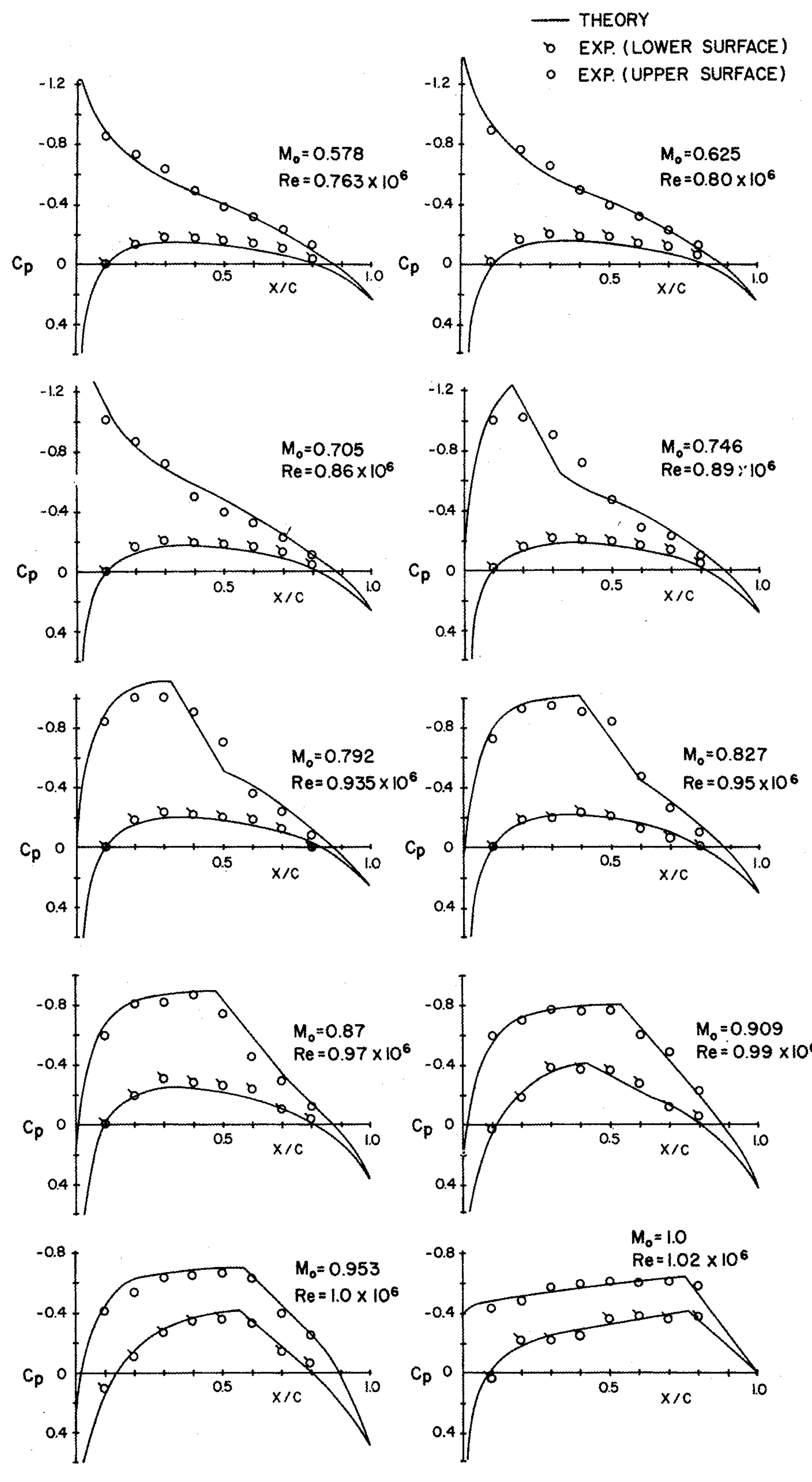

Fig. 2 Comparison of theory and experiment for one airfoil.

results were compared with experimental data obtained on two airfoils in the NCSU Transonic Wind Tunnel. The airfoils are shown in Fig. 1 and the comparison for one of them with theoretical predictions is given in Fig. 2. A summary of lift and moment characteristics is shown in Fig. 3. Note that the predicted pressures are all within 10% of the experimental points. The worst agreement between the prediction and experiment is at Mach numbers just above the critical. This is due to the very diffuse nature of the shock boundary-layer interaction and demonstrates quite explicitly the need to include viscous effects in any prediction scheme—at least when laminar boundary layers are present.

The experimental data were recorded with errors not

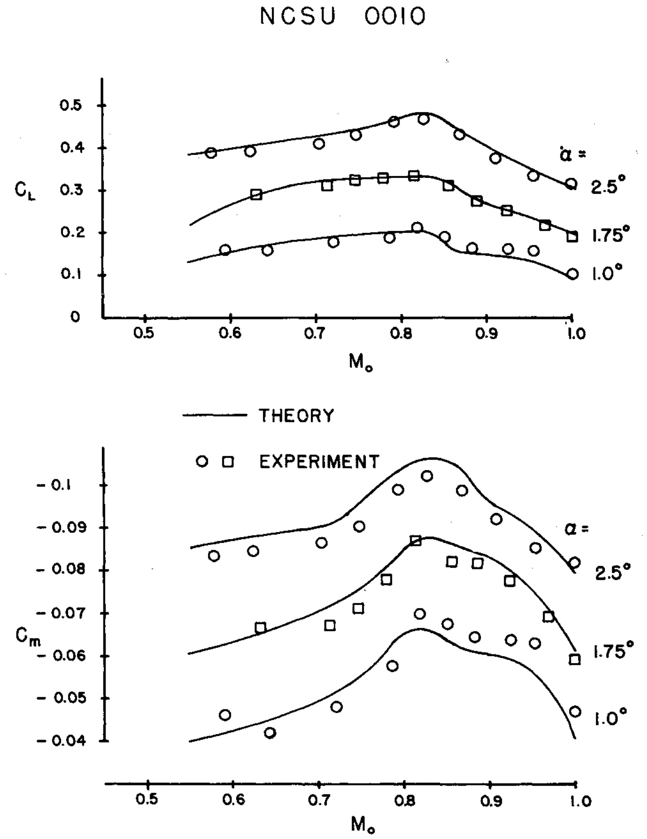

Fig. 3 Summary of results for one airfoil.

larger than 1%. The model consumed about 3.5% of the tunnel cross-sectional area at $\alpha = 0$ and the two slotted walls were 50% open. It was felt that under these conditions wall interference corrections were not necessary and none were made. Model angle of attack was limited to about 4° because increased blockage then limited the attainable Mach number. Generally, the agreement between prediction and experiment deteriorated as α increased. This is thought to be due to the increasing importance of boundary-layer displacement effects associated with the adverse pressure gradient over the rear portions of the airfoil at higher angles of attack. The simple vortex distribution scheme used to compute the $M = 0$ distribution of course does not permit this displacement effect to be considered as does the more inclusive method of Ref. 5.

The success of the simple prediction scheme offers sufficient incentive to warrant the effort to include a better vortex distribution method and a more accurate description of the shock pressure rise in the calculation procedure. If these modifications improve the agreement between prediction and experiment without excessive increases in computer time, it would appear reasonable to attempt to include a more analytical version of the method of Ref. 7 as well.

References

[1]Spreiter, J. R. and Stahara, S. S., "Calculative Techniques for Transonic Flows About Certain Classes of Airfoils and Slender Bodies," NASA CR-1722, April 1971.

[2]Thompson, N. and Wilby, P. G., "Transonic Aerodynamics," *AGARD Conference Proceedings No. 35*, (AD—685270), Sept. 1968.

[3]Fitzhugh, H. A., "Method for Predicting Wing Section Pressure Distributions, Lift, and Drag in Transonic Mixed Flows," *Journal of Aircraft*, Vol. 7, No. 3, May 1970, pp. 277–279.

[4]Weber, J., "The Calculation of the Pressure Distribution Over the Surface of Two-Dimensional and Swept Wings with Symmetrical Aerofoil Sections," R & M 2918, 1956, Aeronautical Research Council, Calgary, Canada.

[5]Stevens, W. A., Goradia, S. H. and Braden, J. A., "Mathematical Model for Two-Dimensional Multi-Component Airfoils in Viscous Flow," NASA CR-1843, July 1971.

[6]Truitt, R. W., "Shockless Transonic Airfoils," AIAA Paper 70-187, New York, 1970.

[7]Sinnott, C. S. and Osborne, J., "Review and Extension of Transonic Aerofoil Theory," R & M 3156, 1958, Aeronautical Research Council, Calgary, Canada.

[8]Murphy, J. D. and Rose, W. C., "Analytical Methods in Aircraft Aerodynamics," NASA SP-228, 1970.

Correlation of Wing-Body Combination Lift Data

Leland M. Nicolai* and Felix Sanchez†
U. S. Air Force Academy, Colo.

WHEN a wing is added to a body at low angles of attack, there are mutual interference effects present between the components that makes the lift of the wing-body combination greater than the sum of the lift of the individual components. These interference effects are 1) the effect of the body upwash or cross flow on the local angle of attack of the wing; 2) the effect of local body-flow parameters such as Mach number and dynamic pressure on the wing characteristics; 3) the effect of the lift carryover from the wing onto the body; 4) the effect of wing upwash on the body ahead of the wing; 5) the effect of the wing lifting vortices on the body behind the wing.

These mutual interference effects on the wing-body lift are generally small for configurations with body diameter to wing span ratios, d/b, less than 0.1 (typical of high aspect ratio aircraft). For d/b ratios greater than 0.1, typical of low aspect ratio aircraft and missiles, the interference effects are significant and should be accounted for in order to properly determine the lift characteristics of particular wing-body configurations. The method of Pitts, Nielsen, and Kaattari[1] considers the aforementioned five interference effects and predicts the lift characteristics of wing-body combinations with an accuracy of ±10%. Their method suggests that the wing-body lift curve slope at all

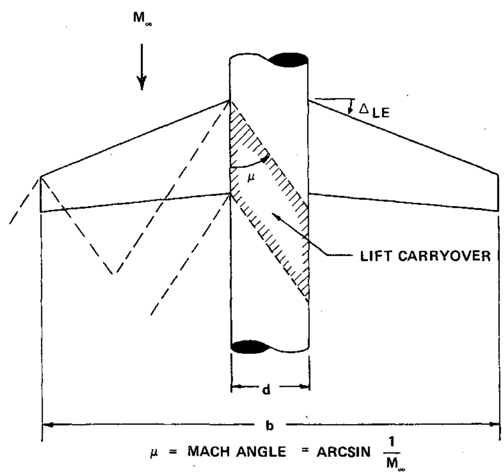

Fig. 1 Sketch of wing-body showing lift carryover region and wing-body parameters.

Received April 24, 1972; revision received November 27, 1972.
Index category: Airplane and Component Aerodynamics.

*Associate Professor of Aeronautics. Member AIAA.
†1st Lieutenant.